Seasat-A Attitude Control System

R. Weiss,* J.J. Rodden,† R.J. Hendricks,‡ and S.W. Beach§
Lockheed Missiles and Space Co., Inc., Sunnyvale, Calif.

The Seasat-A is a new NASA space program using a satellite for making a variety of measurements of the world's oceans. It will map the global ocean geoid; chart ice fields; measure sea surface topography; monitor on a global scale the sea's surface conditions of wave height, direction, and current patterns; and demonstrate an operational global sampling and data processing system. The paper discusses the Agena-based spacecraft attitude control system. The attitude control system operates in four distinct modes: thrust vector control during ascent, mass expulsion maneuvering for post-injection orientation, mass expulsion gyrocompassing with horizon sensor and gyros for attitude hold during orbit velocity adjust, and biased momentum storage with magnetic desaturation on orbit.

Introduction

SEASAT-A is an experimental ocean-survey satellite which is scheduled to be launched in 1978 from Vandenberg Air Force Base, California. Seasat-A will circle the earth 14 times daily and cover 95% of global ocean area every 36 h. It will send back information on surface winds and temperatures, currents, wave heights, ice conditions, ocean topography, and coastal storm activity.

The Seasat-A vehicle will be equipped with five special sensors – three types of radar and two different radiometers – to make its oceanographic measurements. Expected to weigh about 1800 kg (4000 lb), the satellite will be three-axis stabilized and will operate from near-polar orbit at an altitude of 800 km (430 n.mi.).

The Seasat-A attitude control system controls the attitude of the satellite system during injection into final circular orbit after Atlas boost, during orbit adjust and trim phases, and throughout the 3-yr mission. Ascent and injection guidance and attitude control are provided by the Agena spacecraft with a gyrocompassed mass expulsion system. The system can be used as an orbit adjust attitude control system in conjunction with an orbit adjust propulsion system, and as a backup to the magnetic momentum desaturation system. This Agena guidance and control system has evolved over the last 15 years into a versatile injection and orbit control stage used for hundreds of single- and multiple-burn injections into both low- and high-altitude orbits.

After Agena-associated disturbances such as fuel dump and maneuvering to final orbital orientation, stabilization is handed off to the on-orbit momentum and autonomous magnetic system. On-orbit attitude control functions are performed by a system that has its functional roots in the gravity-gradient momentum bias technology used in the STP P71-2 (SESP) spacecraft and the Canadian Communications Technology Satellite. The equipment used to mechanize this biased momentum storage on-orbit system is evolved from the Ithaco, Inc., Nimbus, Landsat, OGO, SAS-3, IUE, and STP 72-2 Scanwheel¶ systems.[1]

Received June 13, 1977; presented as Paper 77-1057 at the AIAA 1977 Guidance and Control Conference, Hollywood, Fla., Aug. 8-10, 1977 (in bound volume of Conference papers); revision received Oct. 3, 1977.

Index categories: Spacecraft Dynamics and Control; Spacecraft Navigation, Guidance and Flight-Path Control; Spacecraft Systems.

*Advanced Systems Staff Engineer. Associate Fellow AIAA.

†Dept. Manager, Guidance and Control Systems. Member AIAA.

‡Group Engineer, WBS Manager, Attitude Control System, Seasat-A.

§Research Specialist, Guidance and Control Systems. Member AIAA.

¶Scanwheel is a registered trademark of Ithaco, Inc.

The on-orbit system stabilizes from disturbances in the handoff operation and thereafter maintains 0.3-deg pitch and roll and 0.75-deg yaw accuracy stabilization throughout its 3-yr mission. Periodically, stabilization reference is returned to the ascent gyros while velocity changes for orbit trim are performed.

Determination of the attitude of Seasat-A is performed on the ground using the pitch and roll attitude signals of the Scanwheels and sun aspect information provided by a pair of sun aspect sensors which are not a part of the orbital attitude control system. Attitude determination system performance is discussed in Ref. 2. A block diagram of the ACS configuration is presented in Fig. 1.

Ascent Attitude Control

Functions and Approach

Critical functional capabilities of the ascent system include 1) attitude and rate capture following separation from the Atlas F booster; 2) orientation for and stabilization during the initial Agena engine burn; 3) stabilization during the coast phase to orbit apogee; 4) attitude hold during the Agena engine burn at apogee; 5) attitude orientation for on-orbit operations; 6) stabilization during deployment of all spacecraft and sensor appendages; and 7) attitude capture and reorientation to the orbit attitude following unplanned attitude motions. The ascent system also provides all ascent function sequencing and engine burn control. Agena hardware is used for the ascent system. A functional block diagram of the ascent and initial acquisition equipment is shown in Fig. 2.

During the ascent phase, the spacecraft is oriented with its longitudinal axis horizontal and equipment section forward, with roll, pitch, and yaw defined in the usual manner as rotations about the velocity vector, negative trajectory plane normal, and local vertical, respectively.

Separation

Ascent control begins with separation from the Atlas F booster. During the 72-s coast phase, the three-axis-stabilized Agena ascent controller arrests the separation rates and provides a 90-deg roll maneuver. The Agena horizon sensor assembly (HSA) then establishes the proper pitch and roll reference for Agena first burn, with yaw reference established by the yaw gyro.

First Burn

The Command Sequencer starts the first Agena engine burn, which lasts for approximately 210 s. An engine turn-off signal is generated by a Digital Velocity Meter. This insures that an exact preprogrammed velocity increment is imparted

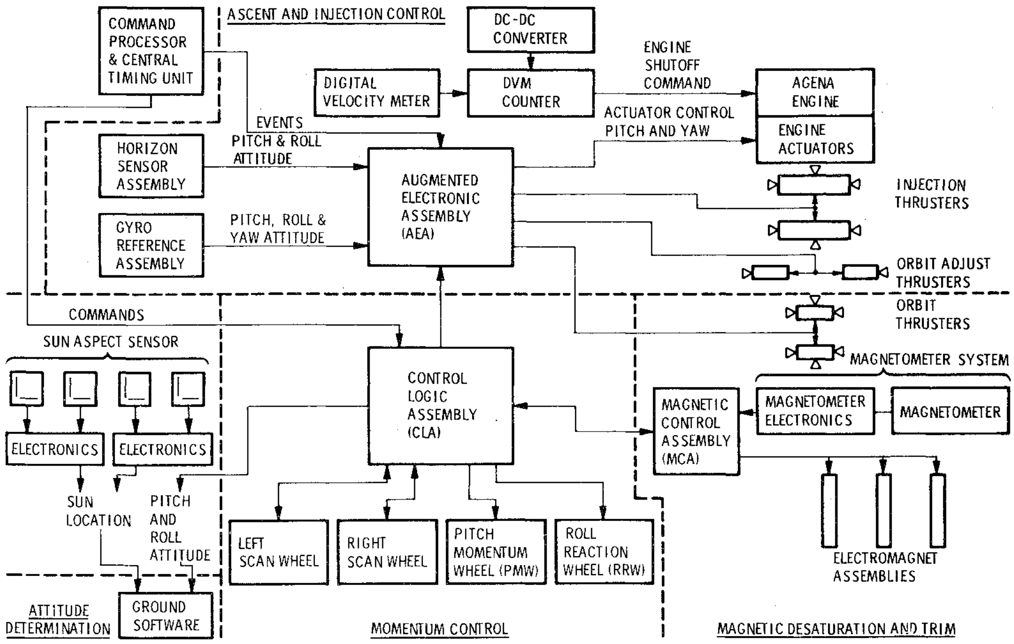

Fig. 1 Seasat-A attitude control system block diagram.

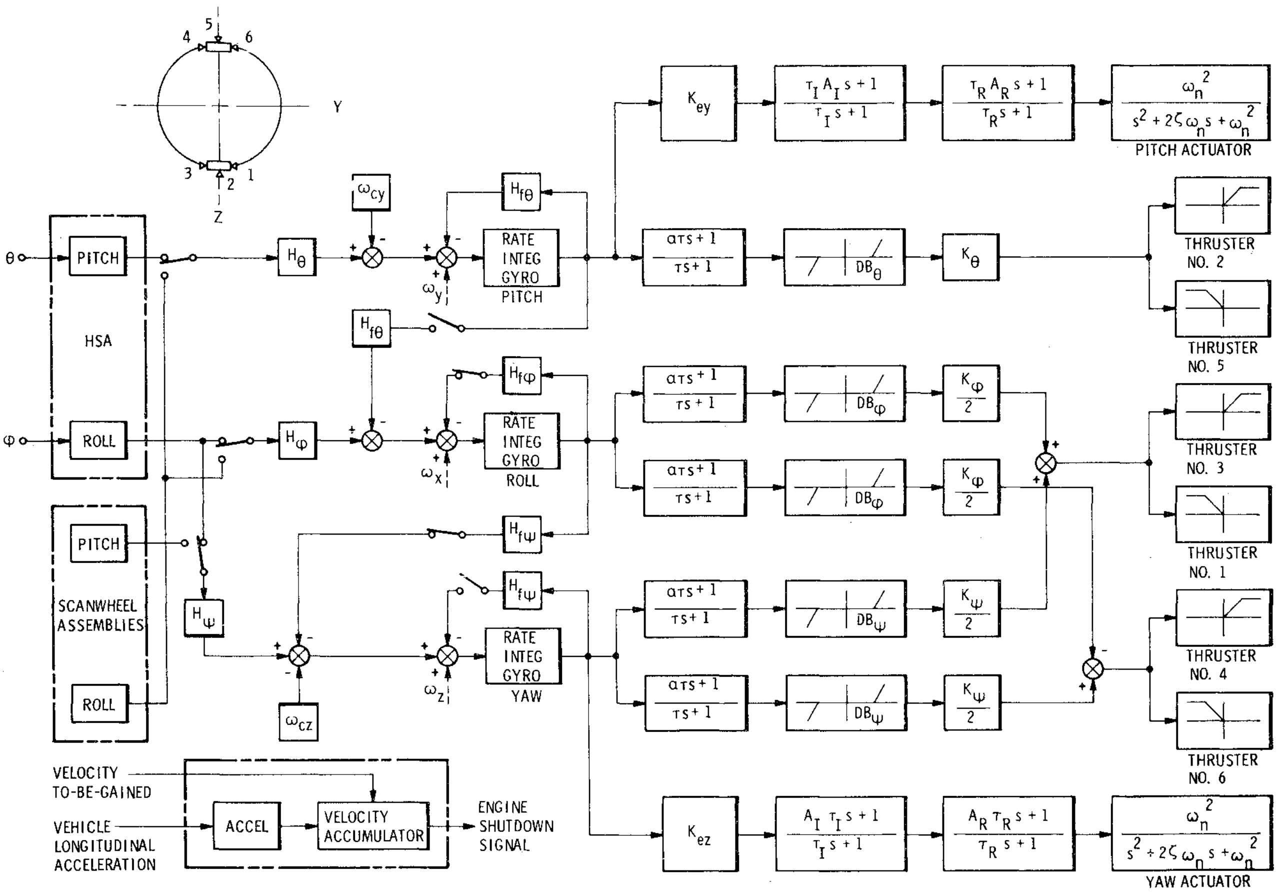

Fig. 2 Ascent control system functional diagram.

by the first burn. Pitch and yaw control torques during the burn are provided by Agena engine thrust vector control; roll is controlled with the high-mode hydrazine thrusters at 53-N (12-lb) thrust levels.

Coast

Between the first and second burns, the vehicle is stabilized with its long axis in a horizontal position by means of the Agena gyrocompass[3] controller. Pitch and roll references are the horizon sensor-updated gyros, and gyrocompass yaw reference is achieved with roll horizon sensor inputs used to update the yaw gyro reference.

Apogee Burn

A second burn of approximately 5-s duration is initiated by the command sequencer at first transfer orbit apogee. Termination is again by means of the Digital Velocity Meter preprogrammed velocity increment. Attitude control is maintained by gyros. Following the burn, the horizon sensor again is added to the control loop and the vehicle reorients to the local horizontal.

Propellant Dump

Following second burn, the remaining primary propulsion propellants are dumped. This requires approximately one-half orbit. During this time, the vehicle is horizontal, with high gain attitude control in the coast gyrocompassing mode. Dumping is activated by the Command Sequencer.

Reorientation

Upon completion of liquid phase propellant dump, the Command Sequencer initiates a 90-deg pitch maneuver at 2.0 deg/s by torquing the pitch gyro. At the completion of the maneuver, the vehicle is oriented with its longitudinal axis pointed toward nadir. At the start of the gyro-referenced maneuver, the ascent horizon sensor signals are removed from the control loop.

A 90-deg yaw maneuver is initiated at the completion of the pitch maneuver in the same manner as pitch, except that the ascent roll gyro is torqued. When completed, the vehicle is oriented in its on-orbit attitude.

Deployment

Immediately following the yaw maneuver, the Scanwheel pitch and roll outputs are connected to the ascent control electronics for gyro drift trim. Payload antennas are deployed upon ground command. A total of 30 min is allotted for this. Control is transferred to the low-mode (2.2-N) thrusters by ground command. The vehicle is again in a gyrocompass mode and stabilized in the on-orbit attitude, with the Scanwheels replacing the ascent horizon sensor function. Scanwheel spinup takes approximately 60 s. Ground commanded deployment of the solar arrays occurs immediately after Scanwheel spinup. Deployment requires from 1 to 5 min. Upon successful deployment, further appendage deployments and attitude configuring are no longer time-critical.

Transfer to On-Orbit Control

Upon completion of all appendage deployments, and monitoring of low mode thrusting indicates that fuel vapor phase venting is complete and that there are no large torques or residual rates on the spacecraft, the momentum control system can be activated by ground command. The procedure for transferring to momentum control is to transfer Scanwheel horizon sensor control from ascent to the on-orbit electronics, and to deactivate the ascent electronics, gyros, and the 2.2-N (0.5-lb) hydrazine thrusters. When attitude transients have died out, as noted on telemetry, the magnetic system is energized for autonomous wheel desaturation. Magnetic trimming is commanded from the ground.

Initial orbit adjust is performed after sufficient ground tracking has occurred to determine ephemeris.

Orbit Attitude Control

Selected Approach

The Seasat-A on-orbit attitude control system is a horizon-referenced momentum control system for pitch and roll control, with pitch momentum bias for gyroscopic yaw control. Damping of the nutation and orbit rate roots is accomplished with an active momentum control system desaturated with an autonomous electromagnetic system.

Hardware Physical Arrangement

The Seasat-A stabilization system is an adaptation of the momentum controller developed and tested for a communication satellite.[4] The system regulates changes of the momentum components in pitch, roll, and yaw in response to horizon sensor-measured pitch and roll signals to maintain pitch and roll attitudes to within ± 0.3 deg (3σ). The yaw axis is restrained by a bias momentum along the negative pitch axis, chosen large enough to keep yaw within limits of ± 0.75 deg. The bias momentum is established by a 20 N-m-s (15 ft-lb-s) momentum wheel aligned in pitch together with the net pitch momentum bias established in a pair of 2 N-m-s (1.5 ft-lb-s) Scanwheels. Changes in pitch momentum are achieved by a $\pm 10\%$ modulation of pitch wheel speed. Yaw nutation damping is provided by differentially changing the speed of the Scanwheels so as to change the yaw components of momentum of the canted Scanwheels. Roll torques are produced by changing the speed of a roll axis reaction wheel. The physical arrangement of control hardware on the Seasat-A spacecraft is depicted in Fig. 3.

Insofar as possible, the system uses current flight-proven hardware. Attitude sensing to within ± 0.20 deg in pitch and roll is accomplished by a pair of infrared horizon sensors incorporated in the Ithaco Scanwheels. The infrared spectral bandpass is between 14 and 16 μm. Running one wheel at its minimum speed, and utilizing constant percentage of peak radiance thresholding techniques, improves the sensing accuracy over earlier versions of this sensor, and makes its performance almost equivalent to the 15-μm horizon sensor analyzed in Ref. 5.

A magnetic torquing system is incorporated to desaturate the momentum wheels. The system comprises three magnetometers to measure the magnetic field, electronic logic and control circuitry, and three electromagnets to torque the vehicle against the Earth's field. The system has three operating modes. In the autonomous mode the coils are energized by internally generated commands in response to tachometer signals to apply the proper torques to desaturate the wheels as necessary. In the ground command mode the required torquing signals are determined by ground analysis based on telemetry data giving vehicle attitude, wheel speeds, and magnetic field vector, and ground commands are sent to the vehicle to energize the magnetic coils. This magnetic trim function will reduce the residual magnetic moment of the spacecraft to below 200 pole-cm in all axes. Reduction of disturbance torques that could be caused by uncompensated magnetic dipoles interacting with the Earth's magnetic field will improve control system performance. Thus, the electromagnetic coils will trim out all steady-state disturbance

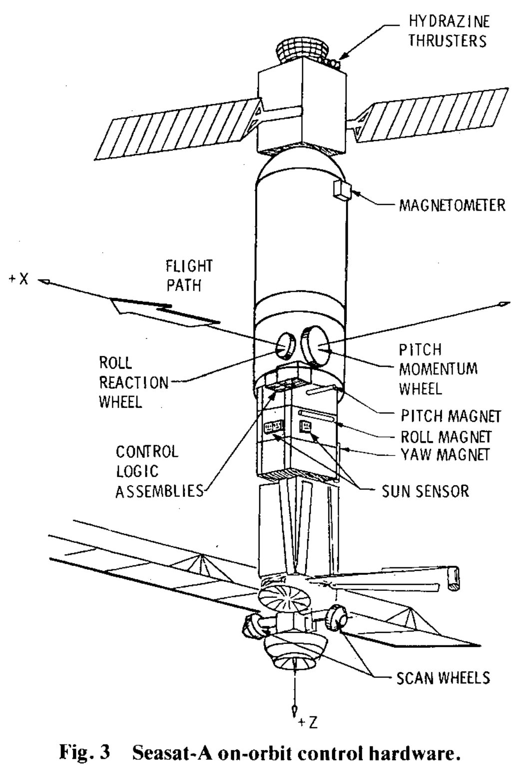

Fig. 3 Seasat-A on-orbit control hardware.

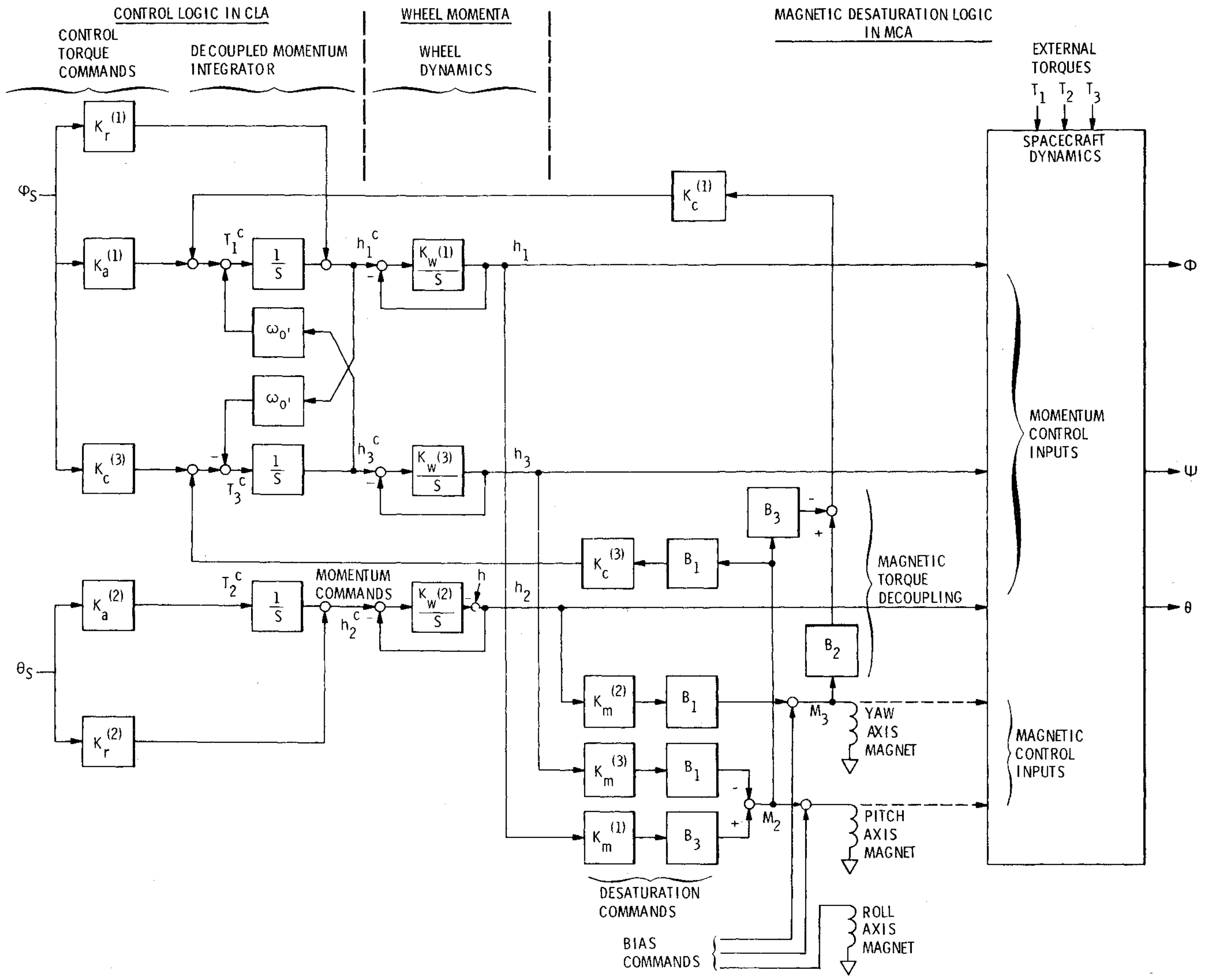

Fig. 4 On-orbit momentum and magnetic control system.

torques produced by the external environment or introduced by misalignments between the spacecraft principal axes and the control axes. The third mode is a backup magnetic control law, which will also be evaluated, that has the potential of magnetically damping nutation of a momentum bias system in case of loss of roll or yaw momentum capability. This backup law is described in Ref. 6.

Pitch Control

The linear pitch equation of motion of the uncontrolled spacecraft in operational notation is

$$[I_2 s^2 + 3\omega_0^2 (I_1 - I_3)]\theta(s) = T_2 - s h_2$$

where I_1, I_2, and I_3 are the roll, pitch, and yaw moments of inertia; $3\omega_0^2(I_1 - I_3)$ is the gravity gradient restoring gain; T_2 is the external pitch torque; and h_2 is the net pitch momentum of the internal wheels.

Pitch control uses conventional reaction stabilization methods. A corrective torque is applied to the pitch momentum wheel drive proportional to measured pitch angle and pitch rate with gains $K_a^{(2)}$, $K_r^{(2)}$, respectively; i.e., pitch control torque $= s h_2 = K_a^{(2)}\theta + K_r^{(2)} s\theta$. Figure 4 is a functional block diagram of the Seasat-A on-orbit momentum and magnetic control system. Values of parameters used in design studies and currently mechanized in the hardware are summarized in Table 1.

When the control relationship is added to the pitch dynamics, the pitch control loop characteristic equation may be expressed as

$$I_2(s^2 + \omega_\theta^2) + K_a^{(2)}\left(\frac{K_r^{(2)}}{K_a^{(2)}} s + 1\right) = 0$$

where ω_θ is the pitch gravity gradient root. The root locus for this equation is shown in Fig. 5.

A small integral term could be used to control the pitch error to null even in the presence of steady pitch gravity-gradient torques. The integral gain is not essential for stable, accurate operation and was omitted since gravity gradient effects can be made sufficiently small by proper inertia distribution. In any case, the accumulation of pitch momentum must be controlled. The total pitch momentum is kept within wheel saturation limits by the magnetic desaturation system.

Roll-Yaw Control

The roll-yaw momentum control system was derived from a concept developed for three-axis stabilization of synchronous communication satellites. The system is a bias momentum controller relying on the gyroscopic effect that tends to align the momentum in the direction of the orbit rate vector. Errors about the pitch and roll axes are controlled actively, but yaw control is achieved through roll-yaw coupling. This coupling

Table 1 Design simulation symbols and values

ϕ_s	HS ROLL SIGNAL	
θ_s	HS PITCH SIGNAL	
h_1, h_2, h_3	WHEEL CONTROL MOMENTUM (ROLL, PITCH, YAW)	
h = 15 FT-LB-S = 22 N-M-S	PITCH BIAS MOMENTUM	
B_1, B_2, B_3	MAGNETOMETER SIGNALS	
M_1, M_2, M_3	CONTROL MAGNET DIPOLES	
$K_a^{(1)}$ = 0.005 FT-LB/DEG	ATTITUDE GAINS	
$K_a^{(2)}$ = 0.02		
$K_a^{(3)}$ = 0.001		
$K_r^{(1)}$ = 1.5 FT-LB-SEC/DEG	ATTITUDE RATE GAINS	
$K_r^{(2)}$ = 1.5		
$K_m^{(1)}$ = 400,000 POLE-CM/FT-LB-SEC OERSTED		MAGNETIC DESATURATION GAINS
$K_m^{(2)}$ = 400,000		
$K_m^{(3)}$ = 400,000 POLE-CM/FT-LB-SEC OERSTED		

provides a gyrocompassing mechanism. The bias momentum is constrained to the local horizontal by the pitch and roll control channels, allowing a weakly constrained oscillatory root in yaw which is damped by the control law compensation. Yaw torque disturbances result in a yaw displacement proportional to the torque and inversely proportional to the gyroscopic stiffness. The steady-state yaw due to torque disturbance, T_3, is given by $\psi_{ss} = T_3/\omega_0 h$. This relationship was used to size the bias momentum based on required yaw accuracy. The wheel system was sized at a bias value of 24 N-m-s (18 ft-lb-s) to limit the disturbance contribution to yaw offset to 0.1 deg for torques up to 42×10^{-6} N-m (31×10^{-6} ft-lb).

It may be noted in the system block diagram of Fig. 4 that electronic differentiation of the roll sensor signal is not used. The control torque is to be integrated in the momentum integrator described below and shown in Figs. 4 and 7. The position signal, weighted with the rate gain, can be added after the roll channel integrator to have the same effect as would an integrated differentiated position signal.

The control law for Seasat-A is an adaptation of one developed for a synchronous satellite.[7] The lower-altitude orbit for Seasat-A and the inertia distribution of a vertical Agena have a significant effect on the roll-yaw control dynamics. The linearized roll-yaw equations of motion are

$$\begin{bmatrix} I_1 s^2 + 4\omega_0^2 (I_2 - I_3) + \omega_0 h & Hs \\ -Hs & I_3 s^2 + \omega_0 h \end{bmatrix} \begin{bmatrix} \phi(s) \\ \psi(s) \end{bmatrix} = \begin{bmatrix} T_1 \\ T_3 \end{bmatrix} - \begin{bmatrix} s & -\omega_0 \\ \omega_0 & s \end{bmatrix} \begin{bmatrix} h_1 \\ h_3 \end{bmatrix}$$

under the assumption that I_1 and I_2 are approximately equal. T_1 and T_3 are external roll and yaw torques on the spacecraft, and the terms at the far right of the torque equations represent the internal disturbances due to momentum changes (sh_1 and sh_3) and gyroscopic torques ($\omega_0 h_3$ and $\omega_0 h_1$).

The characteristic equation of the left-hand matrix is a quadratic in s^2 with separable roots, and can be roughly factored $I_1 I_3 (s^2 + \omega_+^2) \quad (s^2 + \omega_-^2)$, where the lower frequency root ω_- is associated with the orbit rate, and the higher one ω_+ with nutation of a vehicle with a bias momentum. For a synchronous satellite the roots are widely separated, but for a vertical Agena in the Seasat-A orbit the roots are not widely separated. The near-orbit rate root is approximately $1.5\omega_0$, the shift being attributed to the effect of gravity gradient. The nutation root is 3.9 times the orbit rate.

The compensation transfer function developed earlier generated a roll control torque proportional to roll attitude and roll rate with gains $K_a^{(1)}$ and $K_r^{(1)}$, respectively. A portion of this torque was transferred with gain $K_a^{(3)}$ to the negative yaw axis to damp the low frequency orbit rate root.

The modified characteristic equation is then approximately

$$I_1 I_3 (s^2 + \omega_+^2)(s^2 + \omega_-^2) + K_a^{(1)} I_3 \times \left[\left(\frac{K_r^{(1)}}{K_a^{(1)}} s + 1 \right) (s^2 + \omega_0 h / I_3) + \frac{K_a^{(3)} H}{K_a^{(1)} I_3} s \right] = 0$$

A root locus sketch of the modified characteristic is shown in Fig. 6. The nutation complex roots are stabilized directly by the effect of the compensation zero due to the position plus rate feedback gains. The orbit rate roots are stabilized by the location of the gyrocompass zero. The gyrocompass zero is located on a circle of radius $\omega_0 h / I_3$ from the origin. The location on this circle is a direct function of the amount of control torque that is cross-fed into yaw. A damped gyrocompass zero is essential for damping the orbit rate root when wide separation exists in the dynamic roots.

Momentum Integration

An integral part of the control law is the decoupled torque integrator. The basic control law specifies that required control torques be applied to the spacecraft. These torque

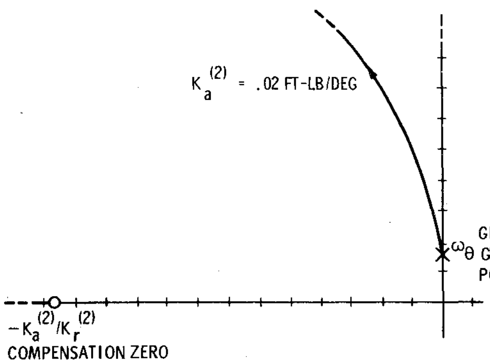

Fig. 5 Root locus for pitch control.

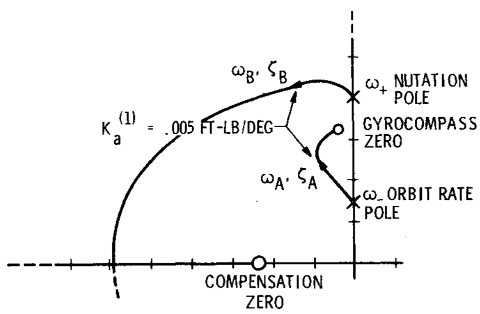

Fig. 6 Root locus for roll-yaw control.

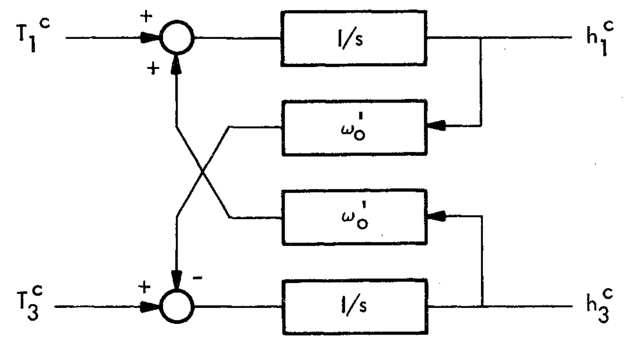

Fig. 7 Decoupled momentum integration.

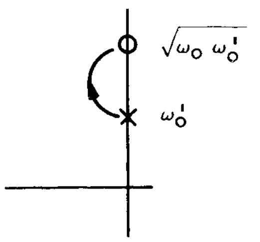

Fig. 8 Root locus for mismatched decoupled integration.

signals are integrated to produce momentum commands for the wheels to follow. The rotation of the spacecraft at orbit rate must be taken into account or the wheel momentum will react gyroscopically with the orbit rate, producing disturbing torques. This effect can virtually be eliminated by decoupling the momentum integration.

Figure 7 is a block diagram of the decoupled momentum integration. In essence the decoupled integration produces momentum commands whose time rate of change with respect to inertial coordinates is the commanded torque input to the integrator.

Any mismatch of the value of orbit rate in the decoupled momentum integrator from the actual value in orbit introduces complex pole-zero roots in the root locus diagram. The transfer function between the physical torque on the vehicle to the commanded value is

$$\frac{T_1^c \text{ actual}}{T_1^c \text{ commanded}} = \frac{s^2 + \omega_0 \omega_0'}{s^2 + (\omega_0')^2}$$

When ω_0 is the actual orbit rate and ω_0' is the mismatched value in the decoupled momentum integrator, the dynamics of such a mismatch is stable (Fig. 8), if the electronic value is below the actual value. In practice it is doubtful whether even an unstable root in the right-half-plane near the imaginary axis would be harmful, but the situation can be avoided in the choice of values for ω_0'. Electronics for this integration were built and used with momentum equipment in an air bearing test and operated satisfactorily.[8]

Magnetic Desaturation

Electromagnets provide desaturation torques to control momentum accumulation without reliance on expendables. Cross-coupling of the desaturation torques occurs that is compensated in the control system without disturbing the vehicle motion. Magnetometer measurements of the local field are used in calculating the desired correction torque and the required compensation. The magnetic control system also generates magnetic moments to trim the unbalanced magnetic moment of the spacecraft.

The desaturation control block diagram was shown in Fig. 4. For roll-yaw desaturation, a magnetic moment M_2 along pitch is made proportional to the excess momentum in the roll-yaw plane and the field strength in that same plane. The yaw axis magnetic moment is proportional to the negative product of the excess pitch momentum and the roll axis field strength. The roll axis magnet is not used for desaturation in order to minimize unwanted magnetic coupling into the yaw control axis, which is the least stiff axis.

The torque (Newton-meters) generated by a vector magnetic moment $\bar{M}$ (Weber-meters) in a vector field B (amperes/meter) is

$$T^m = \bar{M} \times \bar{B} = \begin{bmatrix} +M_2 B_3 & -M_3 B_2 \\ -M_1 B_3 & +M_3 B_2 \\ +M_1 B_2 & -M_2 B_1 \end{bmatrix}$$

Electromagnets aligned with the pitch and yaw vehicle axes are utilized for desaturation of the momentum wheels. To prevent excessive disturbances on the vehicle due to the desaturation magnetic torques, the values for desaturation torques are determined by computing the cross product of commanded magnetic moment and ambient field as sensed by the magnetometers, and the computed torques are applied directly by the wheels to balance the magnetic torque. This accomplishes wheel desaturation directly without the necessity for developing a spacecraft attitude error.

A large yaw disturbance due to magnetic desaturation would be induced if a magnetic moment were applied by the roll magnet to desaturate the pitch wheel. In order to minimize this potential disturbance, only the yaw axis magnet is used to apply torques in pitch.

The control law for desaturation produces magnetic moments M_2 and M_3 along the pitch and yaw axes as functions of excess roll, pitch, and yaw momentum h_1, h_2, and h_3.

$$\begin{bmatrix} M_2 \\ M_3 \end{bmatrix} = \begin{bmatrix} K_m^{(1)} B_3 & 0 & -K_m^{(3)} B_1 \\ 0 & +K_m^{(2)} B_1 & 0 \end{bmatrix} \begin{bmatrix} h_1 \\ h_2 \\ h_3 \end{bmatrix}$$

This desaturation law is attributed to White et al.[9] in recently published surveys of magnetic attitude control systems.[10,11] Substituting this desaturation control law gives the magnetic desaturation torque:

$$\begin{bmatrix} T_1^m \\ T_2^m \\ T_3^m \end{bmatrix} = \begin{bmatrix} -K_m^{(1)} B_3^2 & +K_m^{(2)} B_1 B_2 & K_m^{(3)} B_1 B_3 \\ 0 & -K_m^{(2)} B_1^2 & 0 \\ K_m^{(1)} B_1 B_3 & 0 & -K_m^{(3)} B_1^2 \end{bmatrix} \begin{bmatrix} h_1 \\ h_2 \\ h_3 \end{bmatrix}$$

Note that pitch desaturation causes no yaw disturbance.

Disturbance Torques

The dominant disturbance torques acting on the vehicle are torques due to products of inertia, solar torques, and aerodynamic torques. Anticipated values of inertia products induce bias torques of

$$T_1^i = +4.6 \times 10^{-4} \text{ N-m } (+3.4 \times 10^{-4} \text{ ft-lb})$$

$$T_2^i = -3.8 \times 10^{-4} \text{ N-m } (-2.8 \times 10^{-4} \text{ ft-lb})$$

$$T_3^i = -3.1 \times 10^{-4} \text{N-m} (-2.3 \times 10^{-4} \text{ ft-lb})$$

in roll, pitch, and yaw, respectively. These may be counteracted by effecting wheel momentum biases of −0.47 N-m-s (−0.35 ft-lb-s) on the yaw axis, −0.31 N-m-s (−0.23 ft-lb-s) on the roll axis, and a magnetic dipole moment of 20,000 pole-cm on the yaw axis oscillating at orbital frequency. The yaw momentum and magnetic moment may be accomplished by the pointing control loop with integral gain in roll. The roll momentum bias to balance yaw torque must be commanded (open loop) from attitude determination estimates or previous knowledge of vehicle inertia properties.

Solar torques are expected to be approximately

$$\left.\begin{array}{ll} T_1^s = 0.64\times 10^{-4}\text{N-m} & +0.84\times 10^{-4}\text{N-m} \\ \quad (47\times 10^{-6}\text{ft-lb}) & +\ 62\times 10^{-6}\text{ft-lb} \\ T_2^s = 0.68\times 10^{-4}\text{N-m} & +1.9\times 10^{-4}\text{N-m} \\ \quad (50\times 10^{-6}\text{ ft-lb}) & +141\times 10^{-6}\text{ft-lb} \\ T_3^s = 0.04\times 10^{-4}\text{N-m} & +0.37\times 10^{-4}\text{N-m} \\ \quad (3\times 10^{-6}\text{ft-lb}) & +\ 27\times 10^{-6}\text{ft-lb} \end{array}\right\} \text{at orbital frequency}$$

in roll, pitch, and yaw, respectively. The time histories of solar torques for various sun β angles are shown in Fig. 9. Resulting attitude errors are less than 0.1 deg in all three axes.

Aerodynamic torques may be as large as $T_2^a = 2.3\times 10^{-4}$ N-m (169×10^{-6} ft-lb); $T_3^a = 0.13\times 10^{-4}$N-m($10\times 10^{-6}$ ft-lb), in pitch and yaw, respectively. While uncertainties in aerodynamic torque magnitudes are large because of fluctuations of atmospheric density and varying position of the tracking array, the direction of the pitch torque is determined by the center of pressure location relative to the center of mass, and is known to be opposed to the product of inertia disturbance just discussed.

Torques due to uncompensated residual magnetic dipoles are expected to be small. The ground-commanded magnetic trim capability is capable of trimming out at much as 12,000 pole-cm in 200-pole-cm steps. Typical magnetic dipoles observed on previous spacecraft rarely exceed a few thousand pole-cm.

Simulation Results

Verification of linear analysis results and determination of nonlinear effects were accomplished using an independent computer simulation. The programs developed for this purpose consist of a standardized numerical integration routine, input and output routines, and various specific purpose routines containing coded equations modeling rigid body kinematics and dynamics, orbital motion, atmospheric, solar and magnetic effects, spacecraft flexibility, and typical control system nonlinearities such as deadbands with hysteresis and saturations.

Figure 10 shows the modeled behavior of the system during normal on-orbit operation. Worst case disturbances due to products of inertia, solar torques, aerodynamic torques, and magnetic desaturation are seen to induce yaw errors of less than 0.3 deg. Momentum capability required for the roll and yaw wheels is under 0.7 N-m-s (0.5 ft-lb-s), as are variations required in the pitch wheel momentum. Magnet strengths required for desaturation are under 32,000 pole-cm. The per axis dynamic errors controlling against the worst case torques are therefore kept within 0.3 deg. These errors may be combined with errors from other sources, such as horizon sensing reference error, misalignments, thermal distortions, and desaturation-induced errors, to yield an overall pointing error.

Orbit Adjust and Trim System

The orbit adjust and trim system provides orbit velocity correction and trim following injection of the satellite into orbit and, on ground command, provides for periodic orbital adjust throughout the mission.

The Agena attitude control system used for attitude control and stabilization during Agena ascent and orbit injection provides the required functional capability for stabilization during orbit adjust and trim. Thrusters of the same type are added for orbit adjust velocity increment.

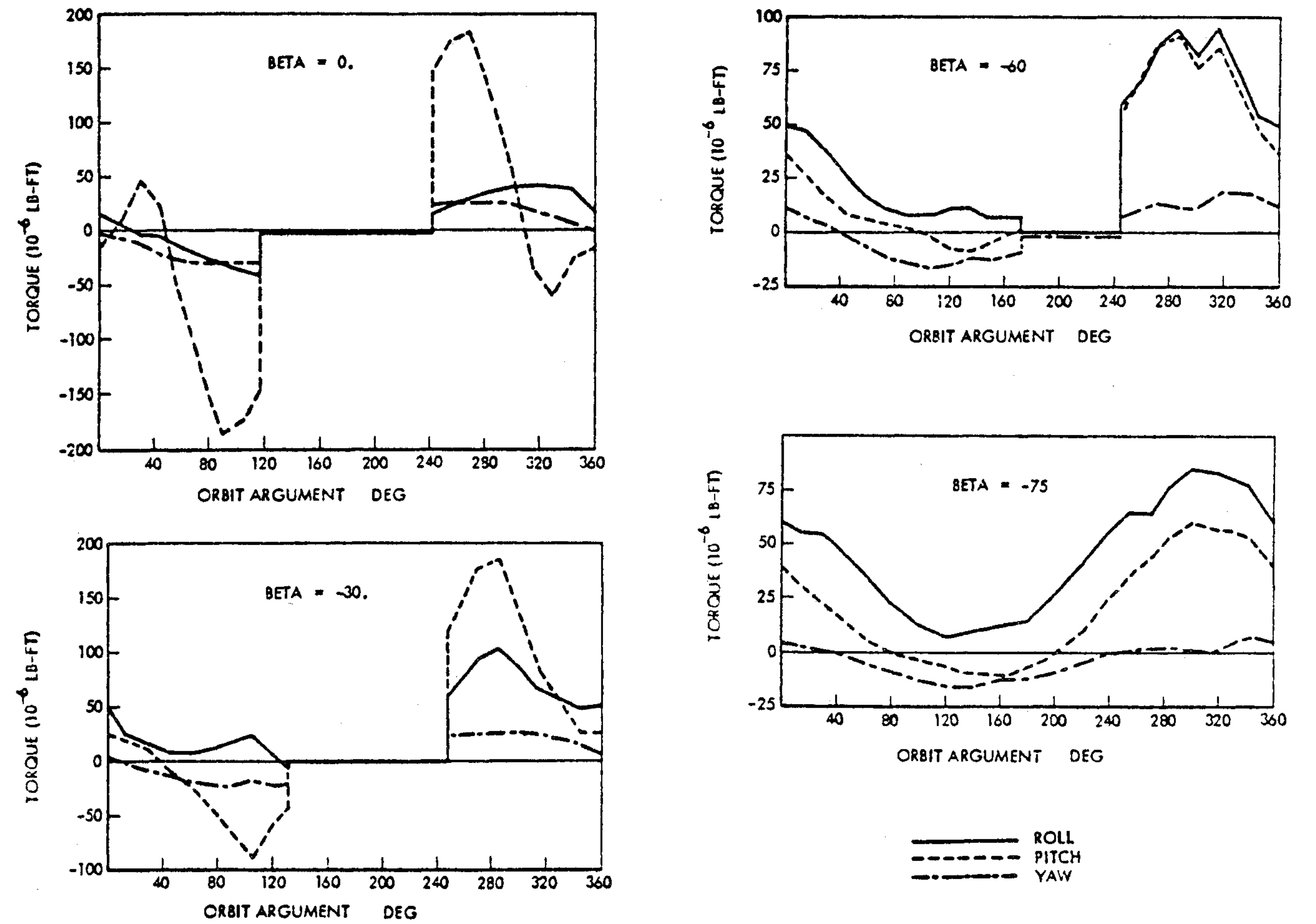

Fig. 9 Solar pressure torques for various sun aspect angles.

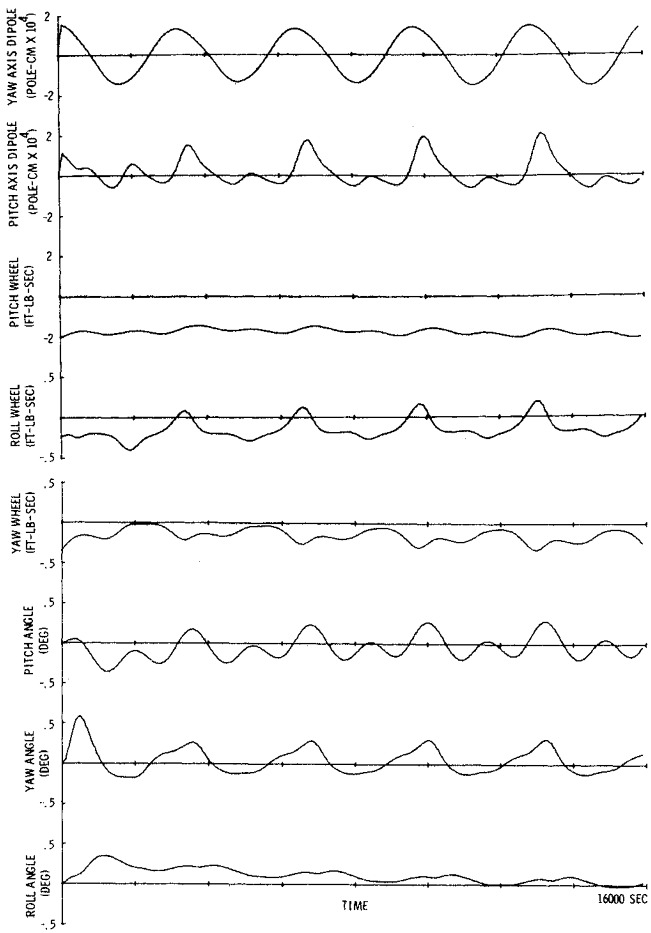

Fig. 10 System behavior during normal orbit operation.

Velocity makeup (ΔV) in the system is provided by two 18-N (4-lb) thrusters aligned to the plus and minus X axes. The proper thruster is fired by real-time ground command over a tracking station or by stored command if the desired point of velocity correction is not in range. Three-axis attitude control is provided by the Agena low mode thrusters (2.2-N), as controlled by the ascent electronics. Attitude reference in pitch and roll is provided by the Scanwheel and the ascent gyro reference assembly (Fig. 2). Yaw attitude reference is provided by the same type of gyrocompassing loops that were used during ascent. Horizon sensor inputs to the ascent electronics are developed in the Scanwheels.

Summary

The attitude control system for Seasat-A has been mechanized using proven equipment in new functional configurations to achieve a low-cost/low-risk solution to the mission requirements. The ascent system is a proven Agena configuration used additionally to provide augmentation to the on-orbit system. The on-orbit control system is an adaptation of the zero-momentum Landsat equipment to a momentum bias configuration. Control laws for the momentum system are adapted from high-altitude mass expulsion control laws for the lower Seasat-A altitude with autonomous magnetic momentum desaturation.

Acknowledgment

This work was performed for the Jet Propulsion Laboratory, California Institute of Technology, sponsored by the National Aeronautics and Space Administration under Contract NAS7-100.

References

[1] "Ithaco Flight History," Ithaco Report SSD-100, Aug. 1977.

[2] Ohtakay, H. and Havens, W.F., "Attitude Determination System for Seasat-A Mission," AIAA Paper 77-1037, Aug. 1977.

[3] Bowers, J.L., Rodden, J.J., Scott,E.D., and DeBra, D.B., "Orbital Gyrocompassing Heading Reference," *Journal of Spacecraft and Rockets,* Vol. 5, Aug. 1968, pp. 903-910.

[4] Dougherty, H.J., Rodden, J.J., Scott, E.D., "Analysis and Design of WHECON – An Attitude Control Concept," AIAA Paper 68-461, San Francisco, Calif., 1968.

[5] Weiss, R., "Sensing Accuracy of a Conical Scan CO_2 Horizon Sensor," *Journal of Spacecraft and Rockets,* Vol. 9, Aug. 1972, pp. 607-612.

[6] Beach, S.W., "Two Momentum Wheel Attitude Control of Gravitationally Stable Satellites," AIAA Paper 76-1949, Aug. 1976.

[7] Lebsock, K.L., Lyons, M.G., and Scott, E.D., "A Double Gimbal Reaction Wheel Attitude Control System for High Altitude Communications Satellites," AIAA Paper 71-949, Hempstead, New York, 1971.

[8] Lebsock, K.L., Flanders, H.R., and Rodden, J.J., "Simulation and Testing of a Double-Gimballed Momentum Wheel Stabilization System for Communication Satellites," *Journal of Spacecraft and Rockets,* Vol. 11, June 1974, pp. 395-400.

[9] White, J.S., Shigamoto, F.H., and Bourquin, K., "Satellite Attitude Control Utilizing the Earth's Magnetic Field," NASA TN-D 1068, Aug. 1961.

[10] Stickler, A.C. and Alfriend, K.T., "An Elementary Magnetic Attitude Control System," AIAA Paper 74-923, Aug. 1974; also *Journal of Spacecraft and Rockets,* Vol. 13, May 1976, pp. 282-287.

[11] Spencer, T.M., "Automatic Magnetic Control of a Momentum-Biased Observatory in Equatorial Orbit," American Astronomical Society Paper 75-031, July 1975.